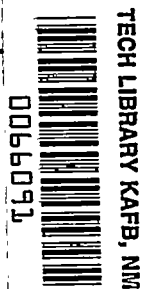


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TECHNICAL NOTE 2990

FLIGHT MEASUREMENTS AND ANALYSIS OF HELICOPTER  
NORMAL LOAD FACTORS IN MANEUVERS

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## SUMMARY

Flight tests have been conducted in two single-rotor helicopters to determine the load factors due to maneuvers. Some additional information has also been obtained from military pilot training and commercial air-mail operations with helicopters.

Load factors of the order of 2.5 were found to be attainable by several different deliberate maneuvers, and this same value was also approached under actual operating conditions. The largest flight loads, as a group, resulted from pull-ups in which both cyclic- and collective-pitch control were applied with suitable phasing.

The assumption that flight load factors are limited to the value that would be computed by assuming all blade sections to be operating at maximum lift coefficient agreed well with flight-test results. This assumption thus provides a convenient method of estimating, for new designs, the maximum obtainable load factors for any given flight condition.

It is concluded that higher speed helicopters and unorthodox configurations may be subjected to load factors materially higher than those experienced by current types.

## INTRODUCTION

The maximum loads attainable in flight are an important factor in the efficient design and safe operation of helicopters, and available evidence indicates that such loads will be increased for newer models capable of higher forward speeds. The possibility also exists that some specialized load-lifter types might experience a decrease in such loads. It is of interest, therefore, to determine the largest load factors actually reached with current types and to correlate these results in a manner permitting examination of future possibilities.

Flight tests for the purpose of determining the load factors resulting from various maneuvers have been conducted jointly by the Civil Aeronautics Administration and the National Advisory Committee for Aeronautics. Two different single-rotor helicopters were used for the tests. The flights were made by CAA pilots; the helicopters were instrumented, maintained, and operated at the Langley Aeronautical Laboratory of the NACA; and the planning of the tests and evaluation of results were handled jointly by the two organizations. Additional data have been obtained, under actual operating conditions, by means of NACA helicopter VGH recorders placed in rotating-wing aircraft used for military pilot training and commercial air-mail operations. Some related information is also available from NACA flying-qualities studies of several different helicopters, one of which was a tandem model. The results are herein analyzed and compared with a simple theoretical method of predicting the maximum load factor that can be attained for a given flight condition.

#### SYMBOLS

|            |   |
|------------|---|
| $g$        | acceleration due to gravity   |
| $n$        | load factor, $\frac{\text{Normal acceleration}}{\text{Gravity acceleration}}$             |
| $\Delta n$ | load-factor increment, $n - 1$  |
| $r$        | radial distance to blade element, ft  |
| $R$        | blade radius, ft  |
| $b$        | number of blades per rotor  |
| $c$        | blade-section chord, ft   |
| $c_e$      | equivalent blade chord (on thrust basis), $\frac{\int_0^R cr^2 dr}{\int_0^R r^2 dr}$ , ft |
| $\sigma$   | rotor solidity, $bc_e/\pi R$  |
| $B$        | tip-loss factor   |

|             |   |
|-------------|---|
| $\alpha$    | rotor angle of attack; angle between flight path and plane perpendicular to axis of no feathering, positive when axis is inclined rearward, radians |
| $\Omega$    | rotor angular velocity, radians/sec   |
| $V$         | true airspeed of helicopter along flight path, fps  |
| $\mu$       | tip-speed ratio, $\frac{V \cos \alpha}{\Omega R}$   |
| $c_l$       | section lift coefficient  |
| $\bar{c}_l$ | rotor-blade mean lift coefficient, $\frac{6C_T}{\sigma} \frac{1}{B^3 + \frac{3}{2} B\mu^2 - \frac{4}{3\pi} \mu^3}$                                  |
| $C_T$       | thrust coefficient, $\frac{T}{\pi R^2 \rho (\Omega R)^2}$   |
| $T$         | rotor thrust  |
| $V_{\max}$  | maximum allowable airspeed  |
| $\gamma$    | mass constant of rotor blade; expresses ratio of air forces to centrifugal forces   |
| $\rho$      | air density   |
| Subscripts: |   |
| max         | maximum   |
| t           | trim  |

#### DESCRIPTION OF HELICOPTERS AND TEST MANEUVERS

Although some information is utilized from flight tests made with other helicopters, the results reported herein are primarily from two series of tests. In each series, a given helicopter was put through specific maneuvers wherein the severity of the maneuver was progressively increased by increasing the magnitude of the control deflection, the rate of deflection, or the time the deflection was held.

### Description of Helicopters

The two helicopters (shown in figs. 1 and 2) used for the flight tests are two-place utility-type aircraft, and both are considered to be reasonably representative of current practice with respect to the major factors expected to affect the obtainable maneuver loads. In particular, both operate at a value of the rotor-blade mean lift coefficient  $\bar{c}_l$  of about 0.45, although at different values of thrust coefficient  $C_T$ . The significance of this parameter  $\bar{c}_l$  with respect to load factors is discussed subsequently in this report, and the relationship between  $\bar{c}_l$  and  $C_T$  is discussed in the appendix.

A comparison of the two helicopters is shown in the following table:

| Characteristic  | Helicopter   |   |
|---|--|---|
|   | A  | B                                       |
| Rotor solidity, $\sigma$ . . . . .                        | 0.033  | 0.06                                    |
| Rotor-blade tip speed, fps . . .                          | 613  | 487                                     |
| Number of blades, $b$ . . . . .                           | 2  | 3                                       |
| Rotor-blade hinge arrangement . . . . .                   | See-saw; blades rigidly interconnected with common flapping hinge and no drag hinges | Each blade has flapping and drag hinges |
| Approximate rotor-blade mass constant, $\gamma$ . . . . . | 5  | 12                                      |

Helicopter A was flown at a gross weight of approximately 2,050 pounds and helicopter B, at a gross weight of about 2,500 pounds.

### Maneuvers Performed

The maneuvers performed include the following:

(1) Jump take-offs, in which the rate of collective-pitch change was varied. "Jump take-off" as used herein means a take-off in which collective pitch is increased at a high enough rate to convert stored

energy in the rotor into temporary added thrust. The rotor is usually overspeeded at low or moderate pitch just prior to such a take-off.

(2) Collective-pitch pull-ups at various airspeeds in both powered and autorotative flight.

(3) Cyclic-pitch pull-ups at various airspeeds in both powered and autorotative flight.

(4) Combined cyclic- and collective-pitch pull-ups in which different combinations and phasing of cyclic- and collective-pitch control were used.

Records of normal acceleration, airspeed, altitude, and control position were obtained by means of standard NACA recording instruments.

The acceleration values given were obtained by fairing through the middle of the vibratory "hash." This fairing is done because the flexibility of these aircraft is too great for motions at this frequency (about 10 cycles per second) to be construed as motions of the entire mass of the helicopter. The typical amplitude of this hash was approximately  $\pm 0.25g$ .

## RESULTS AND DISCUSSION

The maximum load factors (accelerometer readings) and corresponding maneuvers and flight conditions of helicopters A and B are listed in table I. The largest value of acceleration recorded for helicopter A was 2.68g, obtained during a combined cyclic- and collective-pitch pull-up from autorotation. The largest load factor for helicopter B was 2.30, incurred during a cyclic- and collective-pitch pull-up from level flight.

Load factors in maneuvers are influenced not only by the magnitude of control displacement, but also by the rate at which the control is applied. Figure 3 shows the effect of varying the rate of collective-pitch change during jump take-offs in which full travel of the control was used. In figure 4, the results of varying the amount of collective pitch applied at a nearly constant rate (40° per second) are presented for a range of airspeed. Figure 5 illustrates the effect of varying, by suddenly displacing and holding the control, the amount of longitudinal cyclic pitch at various airspeeds.

As part of a general investigation of helicopter flight loads under actual operating conditions, the NACA has also obtained a limited amount of data on military pilot training and commercial air-mail transport.

The largest load factors recorded thus far from these operations have been a gust load factor of 1.88 for the air-mail carriers and a maneuver load factor (landing flare from an autorotative approach) of 2.4 for the pilot training program (incremental accelerations of 0.88g and 1.4g, respectively). Greater sampling may be expected to produce larger values. The helicopters from which these data were obtained are similar to those of the present investigation.

#### Means of Attaining Maximum Flight Loads

Basic methods.- Maneuver loads are basically produced either by cyclic-pitch changes which change the rotor angle of attack or by sudden collective-pitch changes. Figures 6 and 7, respectively, illustrate these cyclic- and collective-pitch maneuvers for three different helicopters. The response of the helicopter is fundamentally different for the two methods of control. For cyclic-pitch changes, the major part of the thrust change develops as a result of angle-of-attack change of the aircraft and, because of fuselage pitching inertia, appreciable time (usually 2 or 3 seconds) is needed to reach the maximum value. For collective-pitch increase, thrust change results directly from increased blade pitch, with no perceptible delay, and then immediately begins to drop off owing to the change of vertical velocity of the helicopter and the corresponding reduction of blade angle of attack, and the reduction of rotational speed.

Combined deflections.- Within the limitations imposed by blade stalling, the accelerations resulting from combined cyclic- and collective-pitch changes tend to be additive, the maximum values occurring when collective pitch is added at the time of peak acceleration due to cyclic pitch (fig. 8(a)). When the controls which produce collective- and cyclic-pitch changes are moved simultaneously, as in figure 8(b), the resulting acceleration does not reach as large a value as when application of the collective pitch is delayed for 1 or 2 seconds. In the present investigation the largest loads, as a group, were found to be the result of cyclic-pitch pull-ups followed a short time later by increased collective pitch. This phasing of control motion would appear to correspond to a maneuver likely to be encountered in actual operation, for example, in landing flares from autorotational approaches or during avoidance of suddenly seen obstacles.

Pull-ups from a diving attitude.- In a cyclic-pitch pull-up from level flight, the resulting upward inclination of the flight path causes a loss in airspeed; also, the change in the direction of gravity force with respect to aircraft axes results in an increase in the pitching velocity required to maintain a given normal acceleration. Both of these changes act to reduce the load factor obtained. With these factors in mind it was thought that a maneuver wherein the helicopter was in a

diving attitude prior to the pull-up would result in smaller nose-up attitudes and thus make high load factors more readily obtainable. Such a maneuver has been suggested as one of the most likely sources of high load factors in actual operations. For example, if a nose-down attitude occurs while the pilot's attention is distracted by navigational duties, then upon noting that the airspeed is in the process of exceeding the red-line value, he may pull up abruptly to prevent further exceeding the placard airspeed. The test maneuvers actually made, although producing high load factors, involved rather improbable nose-down attitudes (see table I). These maneuvers, however, were started at approximately 50 miles per hour, and similar maneuvers started near  $V_{max}$  would involve much smaller attitude changes for a given load factor.

Jump take-offs.- Load factors greater than 2 were obtained in jump take-offs for both helicopters, when unusually rapid deflections were made. Examination of the problem indicates that the attainable load factor would tend to increase in proportion to the square of the rotor rotational speed. Therefore, if, in a particular design, considerable rotor overspeed were allowed in order to permit more effective jump take-off, the resulting increase in attainable load factor should be considered.

Maneuvers in autorotation.- Table I shows that the highest load factors were obtained in autorotation. The placard rotor speed is higher for autorotation than for power-on flight, and the resulting use of higher rotational speeds is felt to be the primary reason for the higher acceleration values. An additional factor is that greater collective-pitch range, from trim to the upper stop, is available in autorotation. Also, in contrast with techniques sometimes employed in autorotative load-factor trials, the engine was left running and hence, because of the action of the synchronizing cam, provided considerable power when the pitch lever was raised. Thus, for maneuvers from autorotation which included use of collective pitch, the rotor speed was higher for this technique than would be true with a stopped engine. Since many landing approaches are made with power settings at or near that for autorotation, but with the engine still operative, the procedure used is considered appropriate.

#### Predictability of Load Factors

Collective-pitch maneuvers.- The acceleration increments resulting from sudden collective-pitch increases at different airspeeds are compared in figure 4 with theoretical values obtained with the aid of figure 1(a) of reference 1. As predicted by the theory, the incremental load factor increases moderately with speed for a given amount of collective-pitch change.

Cyclic-pitch maneuvers.- Prediction of the load factor reached in a cyclic-pitch maneuver is much more involved since the angle-of-attack stability and damping in pitch of the helicopter, as well as its inertia, must be considered. These factors are not known, for the subject helicopters, to a sufficient degree of accuracy to warrant specific comparison with theory. It may be noted, however, that, in figures 7 and 9 of reference 2, reasonable agreement between theory and experiment is indicated for the mild pull-up studied therein. In addition, the experimental trend of load-factor increment with airspeed shown in figure 9 of this paper is in keeping with the theoretical variation with tip-speed ratio of thrust coefficient per unit angle-of-attack change, as shown in figure 1(a) of reference 1; that is, the load-factor increment increases somewhat faster than if it were in direct proportion to the airspeed.

Maximum attainable load factors.- If extremely transient, or vibratory, effects are assumed negligible or left for separate consideration, then the maximum possible rotor thrust cannot exceed the value computed by assuming all blade sections to be operating at  $c_{l_{max}}$ . The maximum load factor attainable will be the ratio of this thrust to the trim thrust, or (approximately) the ratio of  $c_{l_{max}}$  to trim mean lift coefficient  $\bar{c}_{l_t}$ . A more precise relationship is developed in the appendix, where the choice of the value of  $c_{l_{max}}$  is also discussed.

In figure 10, the theoretical maximum load factors for helicopters A and B are shown as a function of the trim mean rotor-blade lift coefficient. For comparison, flight-test values for the more severe maneuvers are also shown in this figure. These curves are based on the following simplified version of formula (A1) of the appendix:

$$n_{max} = \frac{c_{l_{max}} \left( \frac{\cos a_{0n}}{\cos a_{0t}} \right)^3}{\bar{c}_{l_t}} \quad (1)$$

where  $a_0$  is the coning angle and the subscripts  $n$  and  $t$  denote values at time of  $n_{max}$  and at trim condition, respectively.

It should be noted that rotational speed and tip-speed ratio have been assumed constant during the maneuver. The tip-loss factor  $B$ , which is used in computing both  $\bar{c}_l$  and the coning angles, was taken as 0.97.

From examination of figure 10, it is evident that the flight values approach the theoretical maximums in many instances. Load factors calculated on the basis that  $c_{l_{max}}$  is attained on all blade sections

would therefore be a realistic indication of the maximum limits to be expected in flight.

In turn, the use of formulas such as (1) and (A1) for the purpose of estimating the probable effects of changes in design parameters appears justified. For example, the theoretical curves of figure 10 indicate that, for cases where the trim mean lift coefficient is small, there is a possibility of reaching very large accelerations. This parameter  $\bar{c}_{l_t}$  is likely to be small for high-speed helicopters, inasmuch as small values aid in reducing retreating-blade stalling. Conversely, where the trim mean lift coefficient is large, as may be the case for some types of load-lifting, low-speed helicopters, lower maximum acceleration values would be expected.

Change of rotor rotational speed during maneuvers.— The assumption of constant rotational speed used in the comparison just presented warrants some discussion. The results of figure 10 indicate this assumption to be adequate for exploring the effect of design changes on obtainable load factors; apparently, the changes in rotational speed which do occur tend to compensate for the deviations from  $c_{l_{\max}}$  which must surely occur over some portions of the rotor disk. For some purposes, however (including prediction of centrifugal stresses as well as for estimation of load factors for unconventional designs), consideration of rotational-speed change during the maneuver will be important.

The manner in which rotor rotational speed varies during the several types of maneuver may be noted for sample cases in figures 6, 7, and 8. Table II lists the magnitude of the rotational-speed changes recorded for helicopter B. No instantaneous values of rotational speed were obtained for helicopter A, but from estimates based on the vibratory components on various records, the spread of values appears to be similar to that shown for helicopter B; that is, approximately 0- to 8-percent increase.

It should be pointed out that the "trim" rotational speed as used here is the measured value just prior to a specific maneuver. It appears unconservative to assume (as is sometimes done) that in practice the rotational speed at the start of a pull-up maneuver will not be greater than the recommended operating value. In an emergency, or when effecting recovery from the effects of a gust or other inadvertent disturbance, the pilot cannot give his attention to adjustment of rotor speed, nor can he permit rotational-speed changes to dictate his actions. (A pertinent case is the maneuver shown in fig. 3 of ref. 3. In this maneuver, which required unusual corrective action, the rotational speed went above the placard limit. A study of the original records showed that a 13-percent increase in rotational speed occurred in this instance.)

Some additional information is also available from various stability studies (such as those of ref. 3). The rotational-speed changes

during relatively mild maneuvers are found to be much larger per g than for the severe maneuvers of table II; specifically, records of pull-ups and long-period oscillations show typical increases of 2 to 4 percent for  $1/4g$  acceleration increments. The largest increase noted (without change in collective pitch or throttle) was 12 percent; this value occurred during a simulated landing flare with only 0.22g acceleration increment.

In view of the dependence on piloting procedure together with the nonlinear variations of rotational speed with acceleration increment, a statistical approach to the prediction of rotational-speed increase should be appropriate. In turn, it appears that a survey of operating rotational speeds in conjunction with the previously mentioned installations of NACA helicopter VGH recorders would be appropriate.

#### Design Limit Load Factors

Present values.- The most common values currently employed by designers for the positive limit load factor appear to range between 2.5 and 3.0. Values as high as 3.5 and as low as 2.0 are either in use or at least under serious consideration. It is of interest to compare the present results with these values.

As shown in table I, load factors of 2.5 or slightly higher can be attained without undue difficulty by several different test maneuvers. This value of 2.5 has also been closely approached in the actual training operations that have been sampled. Exceeding a value of 3.5 appears to be impossible with typical present-day helicopters unless the placard rotational-speed limit is materially exceeded and improbable unless both rotational speed and airspeed placard limits are exceeded.

Gust loads, incidentally, appear less critical than maneuver loads insofar as present load-factor limits are concerned. In approximately 500 hours of air-mail and pilot training operations, the largest gust load factor, as previously mentioned, was 1.88 (incremental value of 0.88), and calculations indicate that a gust velocity of about 60 feet per second (twice the usual design value) would be required to reach a load factor of 2.5 for helicopter A.

Future trends.- As previously mentioned, high-speed helicopters are likely to operate at materially lower rotor-blade mean lift coefficients in order to avoid stalling; thus, the ratio between maximum and trim values of lift coefficient would be increased and similarly, the available load factor. Also, as speed is increased, a given rotor angle-of-attack change produces a larger thrust increment, so that large load factors may be reached without the large attitude change now involved

in the case of cyclic pull-ups. That unorthodox configurations can cause higher load factors is illustrated by the normal-acceleration value of 4.3g which was reported in reference 4 for a fixed-wing autogyro. Thus, high-speed helicopters and unorthodox configurations may lead to the use of either increased design load factors or some means of artificially limiting maneuver loads.

Artificial limitation of load factors.- Particularly in the case of helicopters which do not require great maneuverability, such as certain cargo and load-lifter designs, it may be possible to effect important savings in structural weight by artificially limiting the load factors which may be produced by pilot action. If such a procedure proves desirable, then from consideration of the present study the following items appear to be among those to be considered:

(a) Automatic regulation of rotor rotational speed, to prevent inadvertently increased values which in turn result in, or at least permit, obtaining greater load factors.

(b) Limits (such as by nonlinear dampers) to the rate of change of collective pitch. If conflict occurred with quick entry into autorotation, this limitation could be applied in the upward direction only.

(c) Introduction of control-force gradients per inch and per g. Such values would be limited by flying-qualities considerations and would perhaps be nonlinear in nature. This procedure assumes incorporation of adequate maneuvering stability, inasmuch as a helicopter lacking in such stability may exhibit dangerous stick-fixed divergences (ref. 3).

Negative load factors.- Some results of deliberate attempts at obtaining negative load factors are reported in reference 5. Pushover maneuvers in a single-rotor helicopter resulted in a minimum acceleration of 0.07g, at which point the blades began hitting the stops. No attempt was made, in the tests reported herein, to produce negative load factors. It is felt that the relatively small negative values in current use are seldom critical in the design of a given part, when all of its functions are considered. Maneuvers producing negative load factors are not likely to be deliberately indulged in, with current helicopter designs at least, because of the likelihood of the blades hitting the fuselage and because control moments are reversed during negative accelerations. (This reversal is postponed somewhat with offset-hinge designs; for example, with a flapping-hinge offset of 3 percent of the blade radius and typical helicopter proportions, the reversal will occur

at a load factor of approximately -0.5.) The presence of little or no negative collective-pitch range further restricts such maneuvers.

While deliberate maneuvers producing negative load factors appear unlikely, the chances of encountering a down gust which could cause an acceleration increment of -1.5g and, hence, a load factor of -0.5 appear about as great as the chances of an up gust causing an increment of 1.5g, since a given negative gust value occurs about as often as a given positive gust value (ref. 6).

Since gusts and inadvertent maneuvers appear to be the more probable sources for negative accelerations, the accumulation of data under actual operating conditions is of particular interest. At the present time, the lowest normal-acceleration values obtained from the previously mentioned installations of NACA helicopter VGH recorders have been 0 and 0.3g (incremental values of -1.0g and -0.7g) for the pilot training and air-mail operations, respectively. Since the sampling represented is extremely limited from a statistical point of view, such installations are being continued.

In contrast to the case of positive accelerations, the limitation on negative load factors provided by the reaching of maximum negative  $c_{l_{\max}}$  on all blade sections is of little practical interest. The stall limitation, for example, may permit accelerations of  $0 \pm 2.5g$  for a case where the requirements call for  $1g \pm 1.5g$  (that is, 2.5g and -0.5g), so that the negative stall limit is 2g beyond the design values considered necessary.

## CONCLUSIONS

On the basis of flight tests of two single-rotor helicopters, plus limited additional data obtained from military pilot training and commercial air-mail operations, the following conclusions are made:

1. The concept that flight load factors are limited to the value that would be computed by assuming all blade sections to be operating at maximum lift coefficient agreed well with flight-tests results. This assumption thus provides a convenient method of estimating, for new designs, the maximum obtainable load factors for any given flight condition.

2. Flight load factors of the order of 2.5 may be obtained by helicopters of the type tested by means of several different maneuvers. This value of 2.5 has also been approached under the actual operating conditions that have been sampled.

3. Future designs capable of higher speeds will tend to experience materially higher design load factors.

4. Within the limitations indicated by conclusion 1, accelerations due to cyclic and collective control are approximately additive if suitably phased. The largest flight loads, as a group, resulted from cyclic pull-ups followed in about 2 seconds by increased collective pitch.

Langley Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., May 14, 1953.

## APPENDIX

## SIGNIFICANCE OF MEAN LIFT COEFFICIENT

## Relation Between Load Factor and Mean Lift Coefficient

In the section on "Maximum Attainable Load Factors," the ratio of  $c_{l_{\max}}$  to  $\bar{c}_{l_t}$  was discussed as an approximation to the maximum attainable load factor. A more precise relation, as derived from the definitions of  $C_T$  and  $\bar{c}_l$  (see section "Symbols") and again if  $\bar{c}_{l_{\max}}$  is taken equal to  $c_{l_{\max}}$ , is

$$n_{\max} = \left( \frac{c_{l_{\max}}}{\bar{c}_{l_t}} \right) \frac{B^3 + \frac{3}{2} B \mu_n^2 - \frac{4}{3\pi} \mu_n^3}{B^3 + \frac{3}{2} B \mu_t^2 - \frac{4}{3\pi} \mu_t^3} \left( \frac{\Omega_n}{\Omega_t} \right)^2 \left( \frac{\cos a_{0n}}{\cos a_{0t}} \right)^3 \quad (A1)$$

where

$a_0$  coning angle

subscripts:

$n$  values at time of  $n_{\max}$

$t$  values at trim condition

The ratio involving  $\mu$  is significant only for cases involving both high tip-speed ratios and material change in tip-speed ratio during the maneuver and is essentially equal to unity for the present study. The term involving coning angles is of rather secondary importance in typical cases, but the trim value of  $a_0$  is usually known and can in any case be readily estimated, whereas the value at the time of  $n_{\max}$  is, for practical purposes,

$$a_{0n} = \frac{c_{l_{\max}}}{\bar{c}_{l_t}} a_{0t}$$

so that this term offers no difficulty. For a first approximation for conventional designs, the remaining term (that involving the ratio of rotational speeds) can be taken as unity; the possibility of significant rotational-speed changes has already been discussed.

A sample case with values well within the range of current practice is as follows: The value of tip-speed ratio  $\mu$  is taken to be 0.25, and the change during the maneuver is assumed to be less than 0.05. If, in addition, the following values are assumed:

$$\bar{c}_{l_t} = 0.45$$

$$c_{l_{\max}} = 1.2$$

$$a_{0_t} = 5^\circ$$

then, the coning angle at  $n_{\max}$  is

$$a_{0_n} = \frac{1.2}{0.45} 5^\circ = 13.3^\circ$$

therefore,

$$n_{\max} = \frac{1.2}{0.45} (1)(1)^2 \left( \frac{\cos 13.3^\circ}{\cos 5^\circ} \right)^3 = 2.67 (0.93) = 2.5$$

#### Relation Between $\bar{c}_l$ and $C_T$

Since many designers may be more accustomed to thinking in terms of values of thrust coefficient than values of mean lift coefficient, it may be well to point out that in spite of the rather lengthy definition of  $\bar{c}_l$  it can be mentally estimated with fair accuracy, for conventional designs at least, from the value of  $C_T$ . For example, for  $\mu = 0.25$  and  $B = 0.97$ ,

$$\begin{aligned}\bar{c}_l &= \frac{6C_T}{\sigma} \frac{1}{0.913 + 0.091 - 0.006} \\ &= \frac{6C_T}{\sigma} \frac{1}{0.998}\end{aligned}$$

so that a negligible error would result from the use of  $6C_T/\sigma$  without the  $B$  and  $\mu$  terms. For hovering, and again for  $\mu = 0.4$ , the error becomes about 10 percent, which may be acceptable for a rapid estimation. Thus, for the frequently used value of  $\sigma$  of 0.06,  $\bar{c}_l \approx 100C_T$ , for tip-speed ratios from 0 to 0.4.

#### Choice of Value for $c_{l_{\max}}$

The theory-experiment comparison presented in figure 10 required a choice of values for section  $c_{l_{\max}}$  for use with equation (1). Reference 7 contains data on section characteristics of practical-construction rotor-blade sections, as obtained in static two-dimensional wind-tunnel tests. From a study of these data a value of  $c_{l_{\max}}$  of 1.2 was chosen as being reasonably representative for the actual blade profiles of both of the test helicopters.

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TABLE I

MAXIMUM LOAD FACTORS AND CORRESPONDING  
FLIGHT CONDITIONS

| Maneuver   | Maximum<br>load factor | Flight condition   |
|--|------------------------|--|
| Helicopter A                                     |                        |  |
| Jump take-off                                    | 2.16                   | -----  |
| Cyclic-pitch pull-up                             | 2.38                   | Level flight at 85 mph   |
| Cyclic-pitch pull-up                             | 2.55                   | Autorotation at 80 mph   |
| Cyclic-pitch pull-up                             | 2.52                   | 50° dive in autorotation;<br>maximum airspeed, approxi-<br>mately 75 mph                           |
| Collective-pitch pull-up                         | 2.18                   | Level flight at 90 mph   |
| Collective-pitch pull-up                         | 2.60                   | Autorotation at 80 mph   |
| Combined cyclic- and<br>collective-pitch pull-up | 2.18                   | Level flight at 50 mph; col-<br>lective pitch applied about<br>$1\frac{1}{2}$ seconds after cyclic |
| Combined cyclic- and<br>collective-pitch pull-up | 2.68                   | Autorotation at 50 mph; col-<br>lective pitch applied about<br>$1\frac{1}{2}$ seconds after cyclic |
| Helicopter B                                     |                        |  |
| Jump take-off                                    | 2.22                   | -----  |
| Cyclic-pitch pull-up                             | 1.90                   | Level flight at 85 mph   |
| Collective-pitch pull-up                         | 1.93                   | Level flight at 50 mph   |
| Combined cyclic- and<br>collective-pitch pull-up | 2.30                   | Level flight at 65 mph   |



TABLE II  
 ROTOR-SPEED INCREASE AT MAXIMUM LOAD FACTOR

[Helicopter B]

| Airspeed,<br>mph                                   | Maneuver  | Maximum load<br>factor reached                                       | $\frac{\Omega_n}{\Omega_t}$  |
|--|---|--|--|
| 6<br>11<br>0                                       | Jump take-off                                     | 1.52<br>1.52<br>2.22   | 1.04<br>1.05<br>1.03   |
| 50   | Collective-pitch pull-up                          | 1.93   | 1.00   |
| 51   | Collective-pitch pull-up<br>in autorotation       | 1.56   | 1.08   |
| 45<br>46<br>47<br>68<br>88<br>70<br>87<br>85<br>85 | Cyclic-pitch pull-up                              | 1.22<br>1.23<br>1.30<br>1.40<br>1.50<br>1.55<br>1.55<br>1.55<br>1.90 | 1.02<br>1.02<br>1.01<br>1.02<br>1.03<br>1.04<br>1.01<br>1.02<br>1.08 |
| 41<br>46<br>52<br>57<br>64                         | Combined cyclic- and<br>collective-pitch pull-ups | 1.80<br>1.95<br>2.00<br>2.04<br>2.30                                 | 1.01<br>1.04<br>1.02<br>1.05<br>1.04                                 |



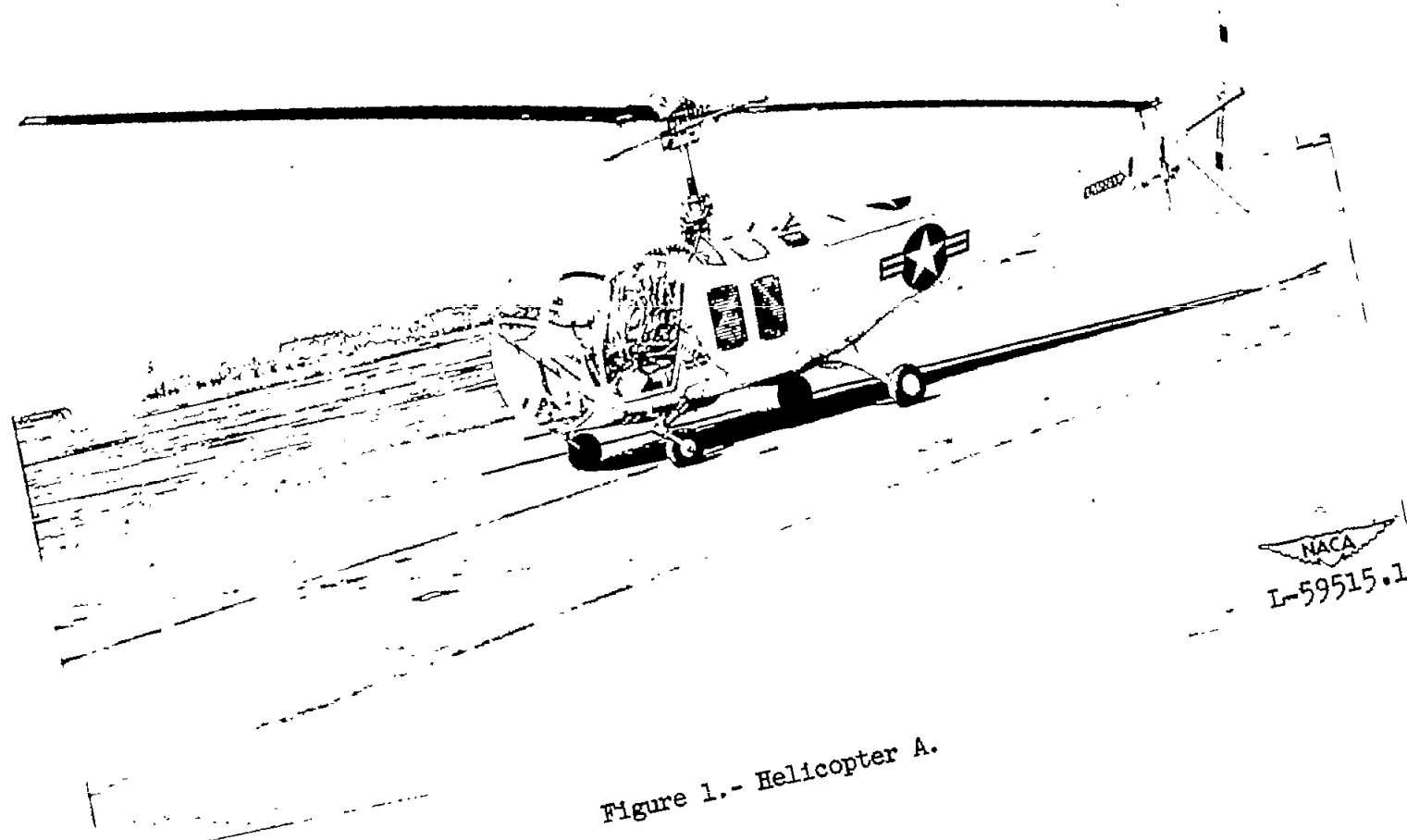


Figure 1.- Helicopter A.

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NACA TN 2990

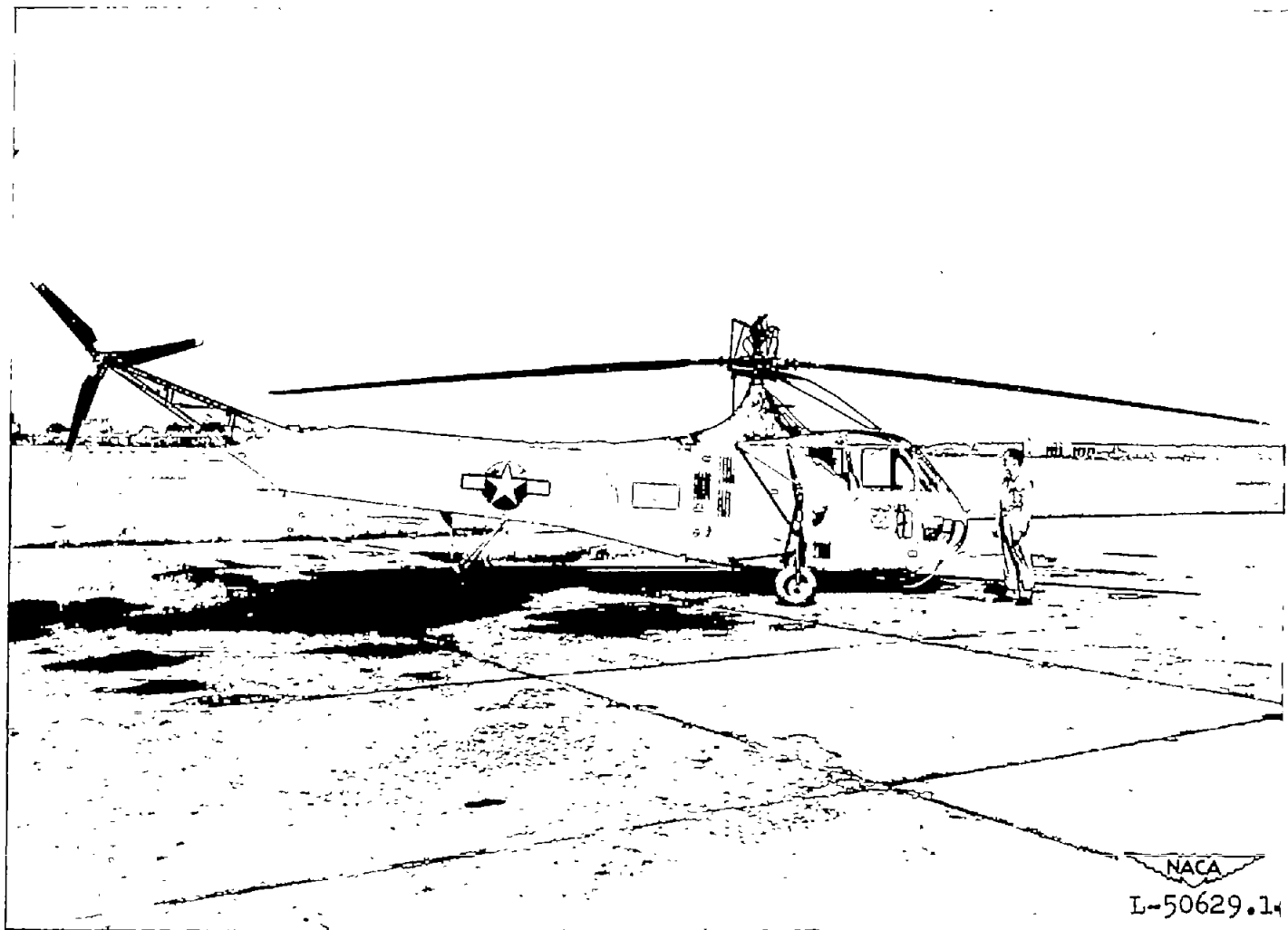


Figure 2.- Helicopter B.

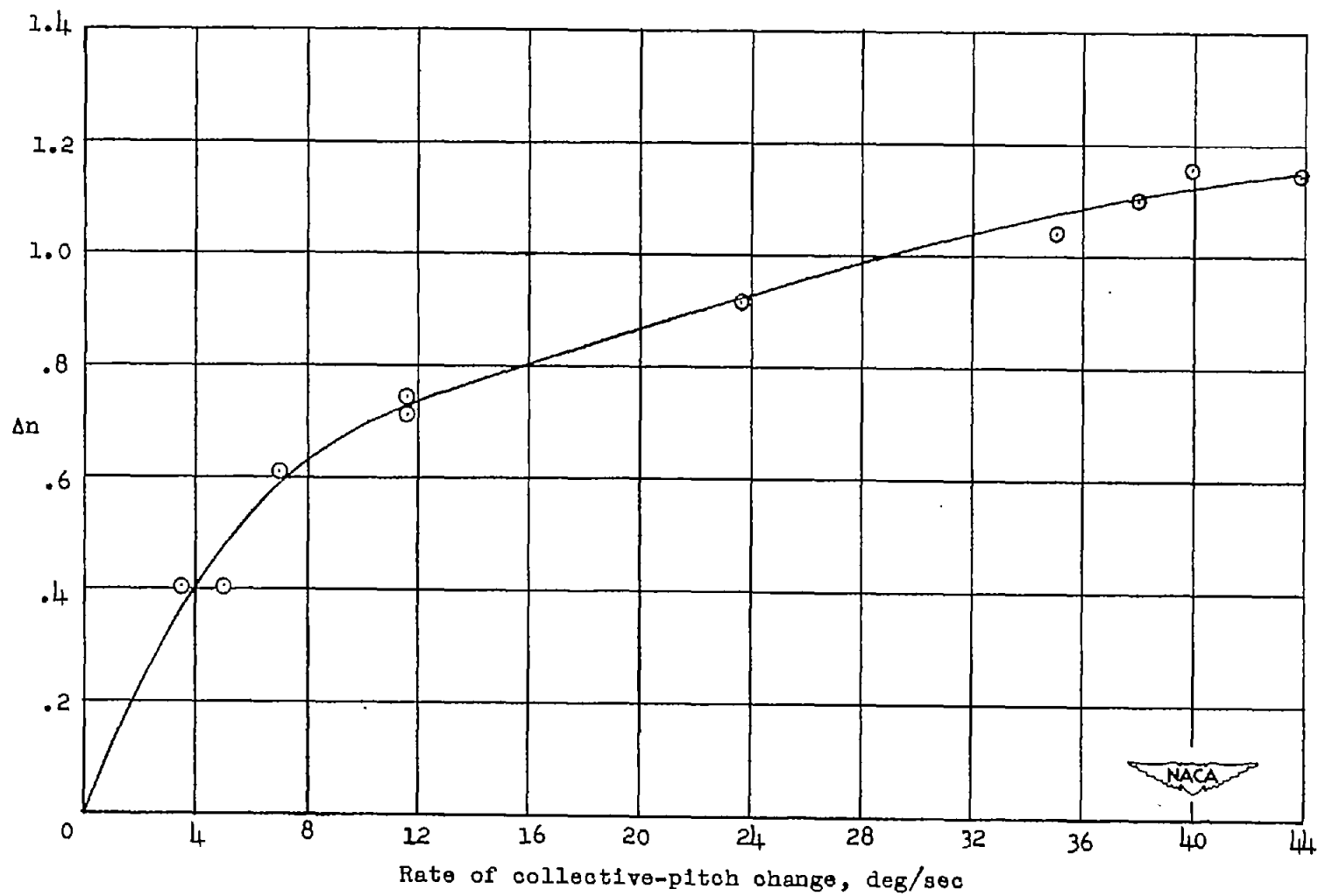


Figure 3.- Variation of load-factor increment with rate of collective-pitch change during jump take-offs using full pitch travel.  
Helicopter A.

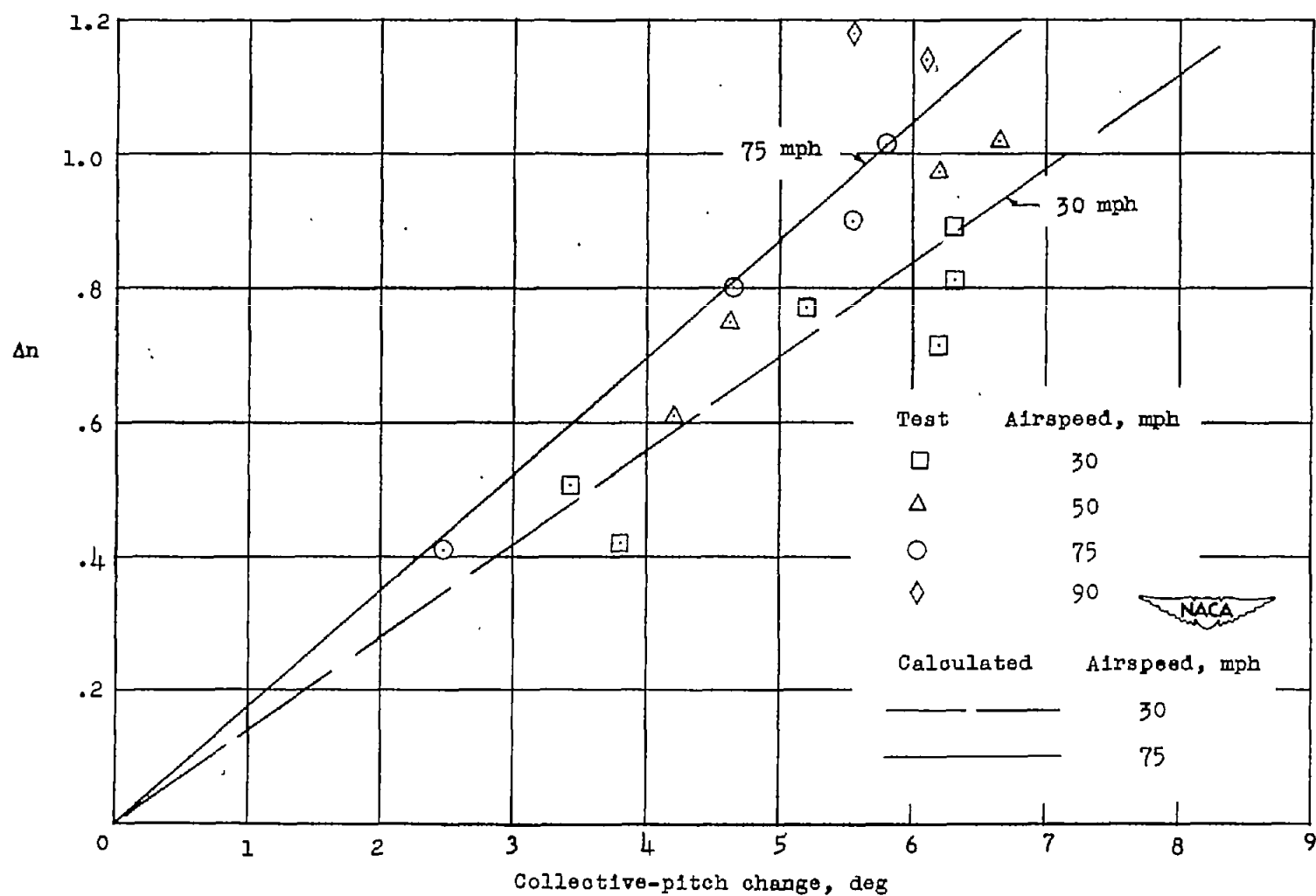


Figure 4.- Load-factor increments due to collective-pitch change in forward flight. Rate of pitch change approximately  $40^\circ$  per second. Helicopter A.

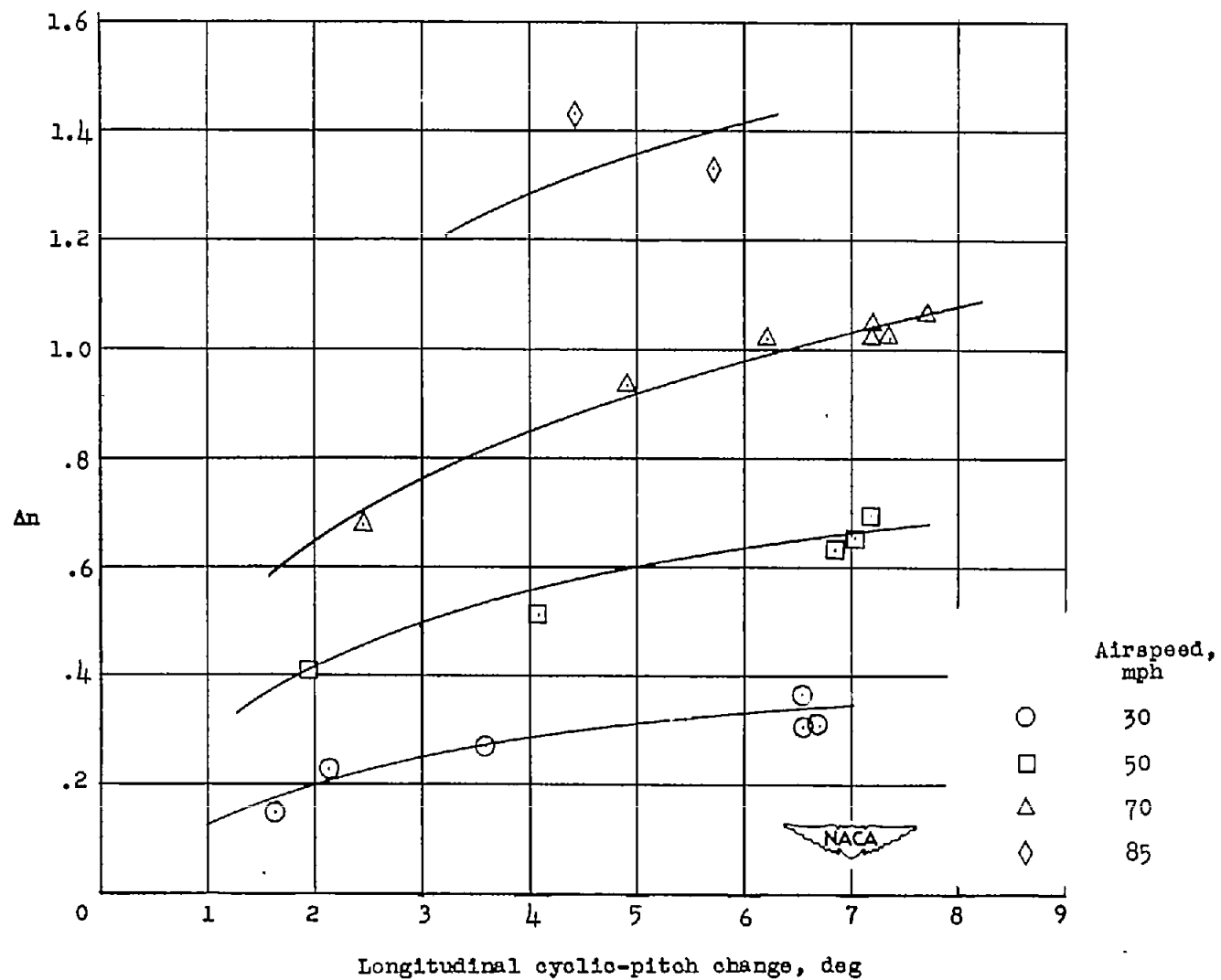


Figure 5.- Load-factor increments due to longitudinal cyclic-pitch change.  
Helicopter A.

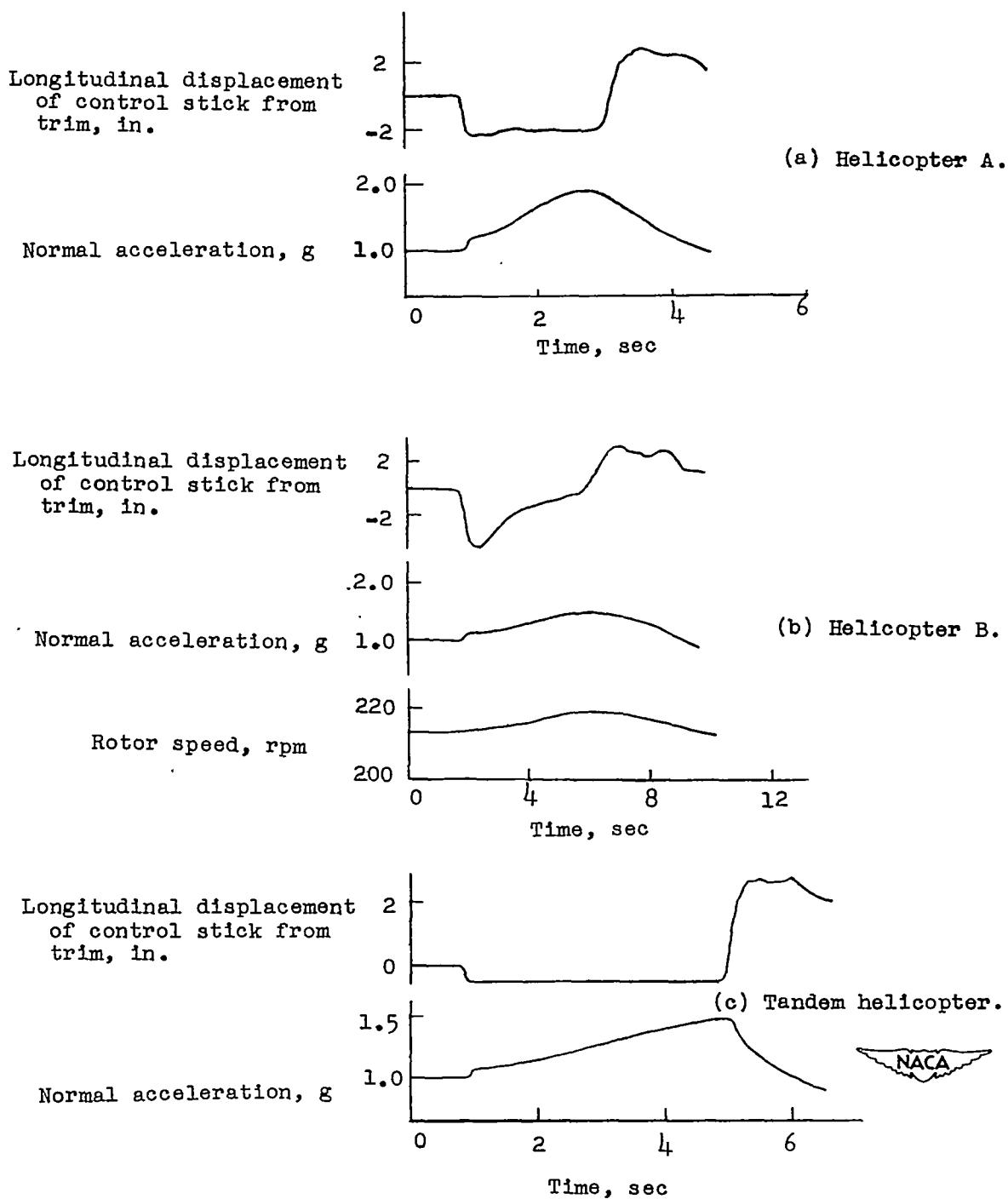


Figure 6.- Time histories of typical cyclic-pitch pull-ups for three helicopters.

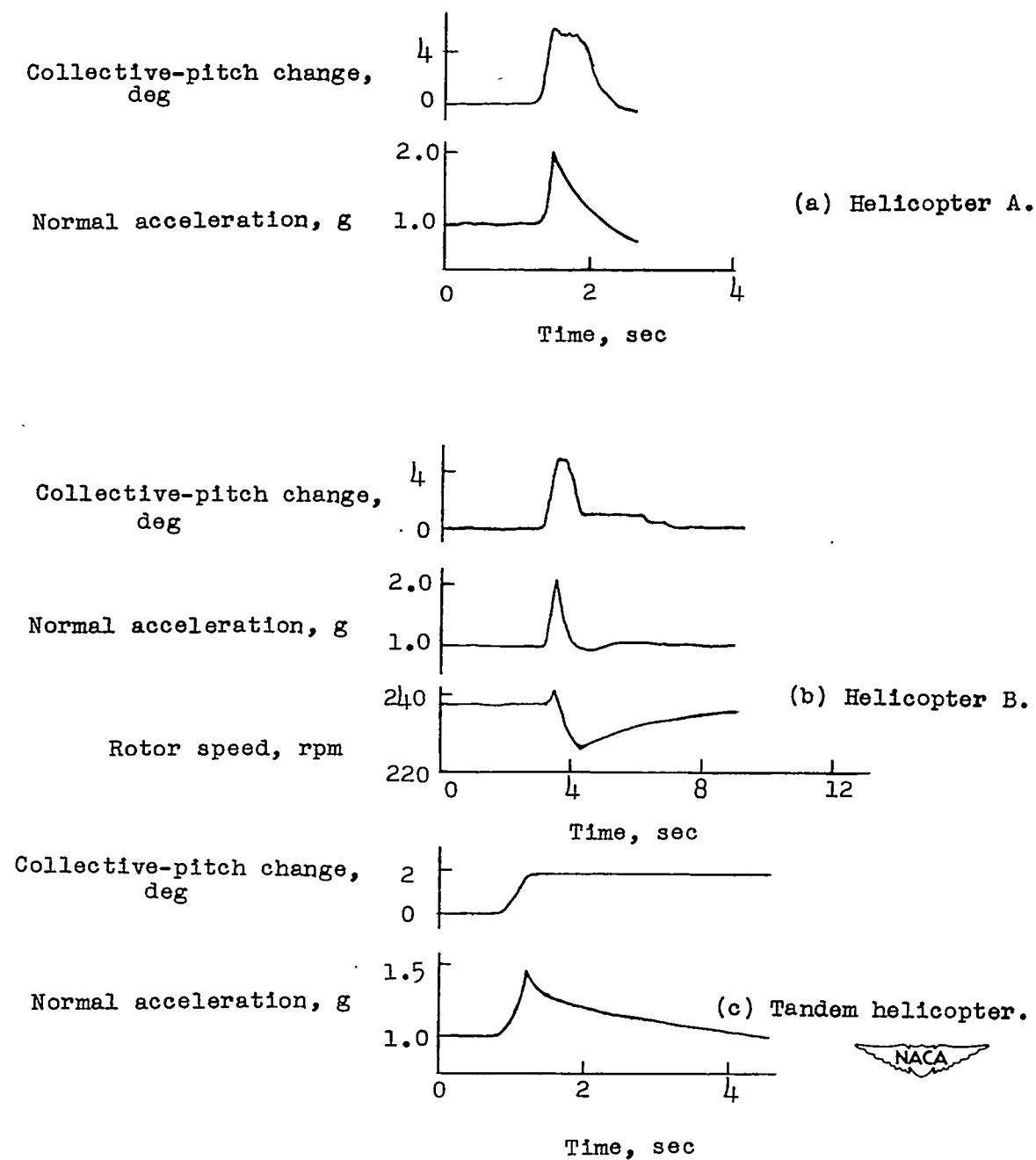
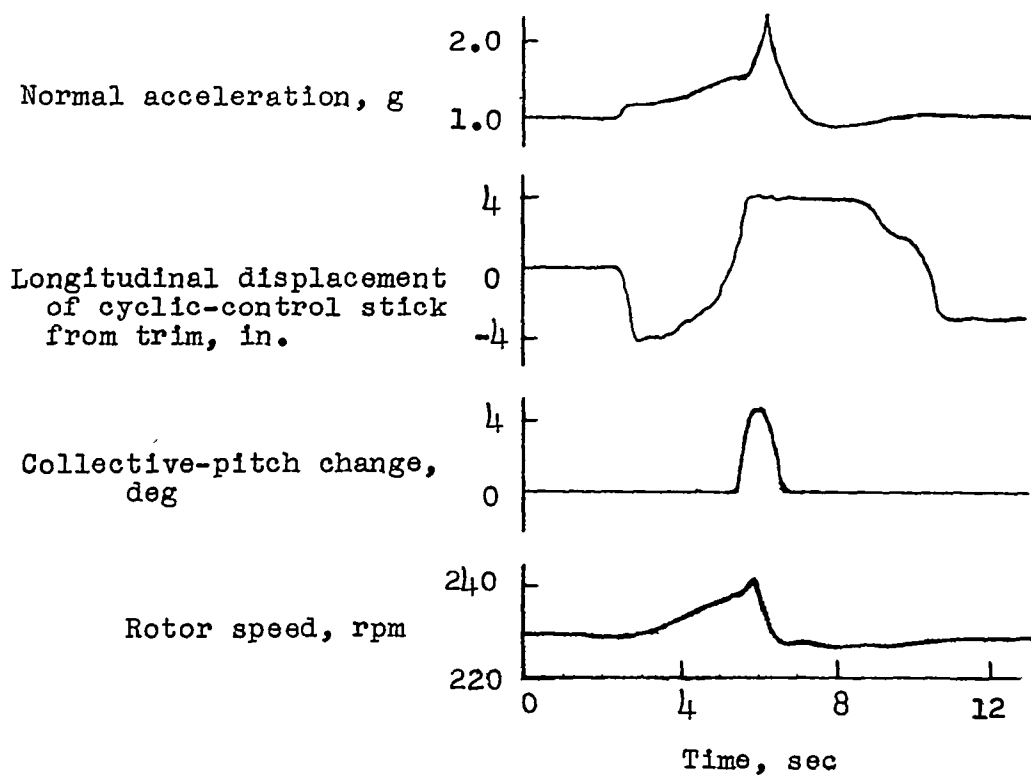
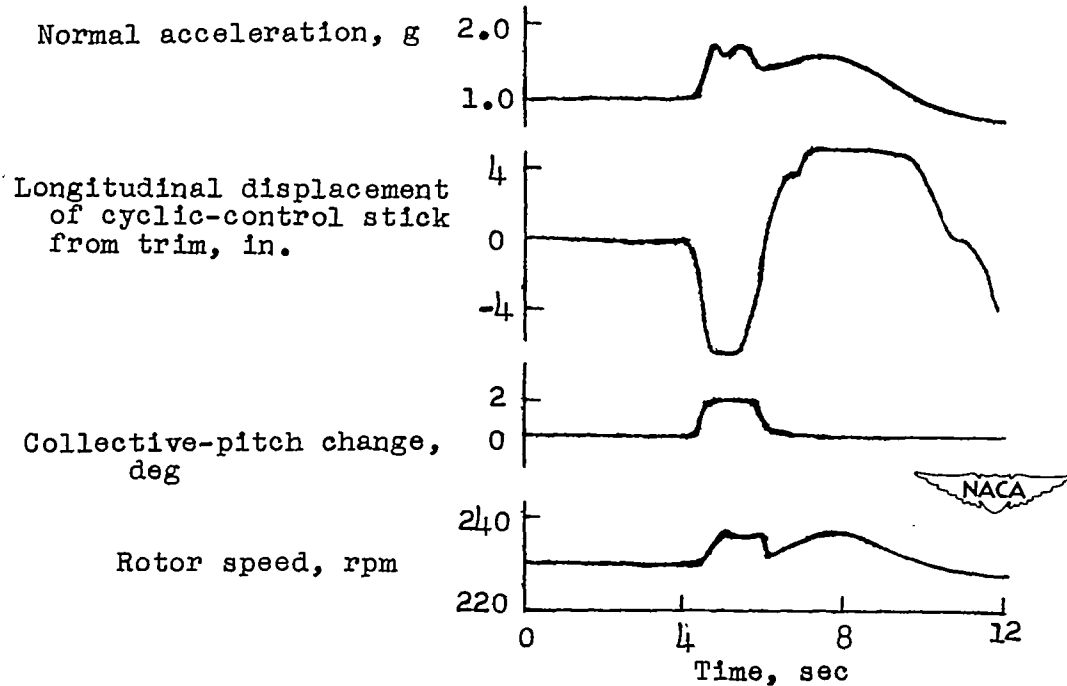


Figure 7.- Time histories of typical collective-pitch pull-ups for three helicopters.



(a) Longitudinal cyclic pitch followed by collective pitch.



(b) Simultaneous cyclic-pitch and collective-pitch change.

Figure 8.- Time histories of two pull-ups using combined cyclic and collective pitch. Helicopter B.

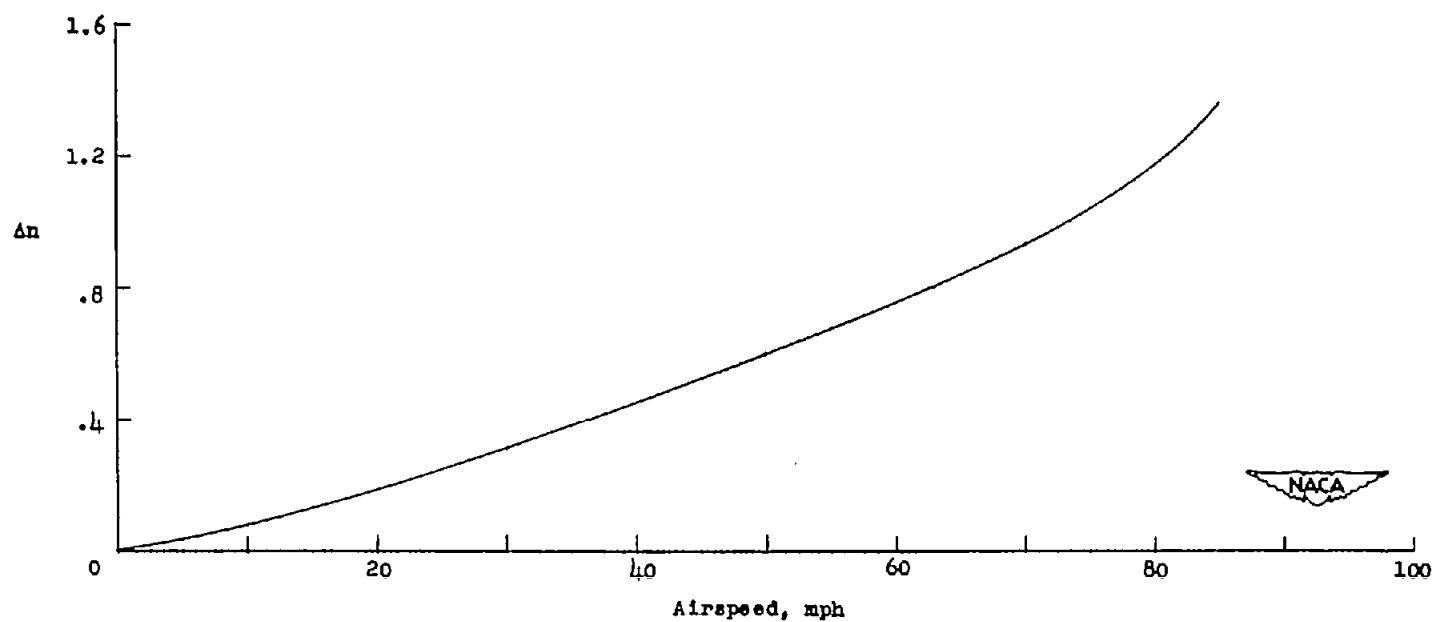


Figure 9.- Experimental variation (obtained from fig. 5) of load-factor increment with speed for longitudinal cyclic-pitch change of  $5^\circ$ . Helicopter A.

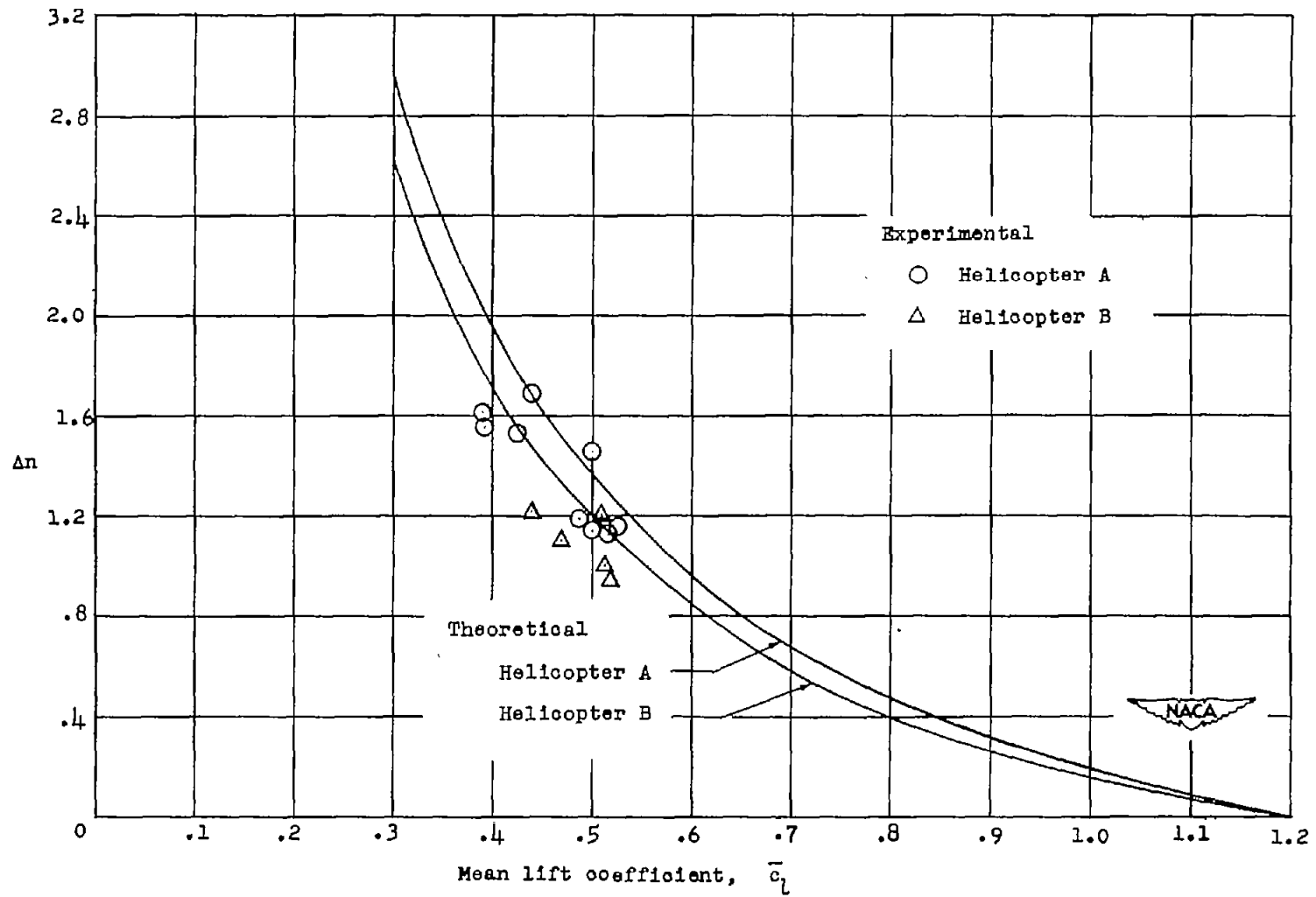


Figure 10.- Comparison of theoretical and flight-test values of maximum load factor as a function of rotor-blade mean lift coefficient at start of maneuver.